BELLCOMM, INC.
1100 Seventeenth Street, N.W. Washington, D.C. 20036

SUBJECT: Some Effects of Decreasing the Launch Window Duration for Apollo Lunar Missions - Case 310

DATE: December 26, 1967

FROM: T. B. Hoekstra

ABSTRACT

Launching an Apollo lunar mission with a predetermined, fixed launch azimuth ("launch-on-time") or at least reducing the duration of the launch window has been suggested as a means of increasing the launch vehicle payload capability and reducing the complexity of preplanning and carrying out a mission. This memorandum presents an analysis of the launch vehicle payload capability, launch window duration, communications and tracking coverage, and abort recovery capabilities as a function of the specific azimuth or range of azimuths chosen for a possible launch. This information is useful in determining the launch azimuth (or range of azimuths) to be used if the launch window is to be reduced.

If the "launch-on-time" concept were employed utilizing a 90° launch azimuth, the launch vehicle payload capability would be increased by over 900 pounds which would allow about a 450 pound increase in the spacecraft inert weight. The number of Apollo/Range Instrumentation Aircraft (A/RIA) could be reduced by four (from eight to four) and the number of abort recovery aircraft could be reduced by one (from six to five). The required number of tracking ships (two; one insertion, one injection) would not be reduced but the number of abort recovery ships could potentially be reduced by two (from seven to five). If 96° (instead of 90°) were chosen as the "launch-on-time" azimuth, the payload capability would only increase about 800 pounds but the Antigua tracking station could provide one minute of post-insertion tracking, perhaps sufficient to eliminate the need for the insertion tracking ship. Economies other than launch vehicle payload and insertion tracking are relatively insensitive to the particular "launch-on-time" azimuth chosen.

(NASA-CR-93077) SOME EFFECTS OF DECREASING THE LAUNCH WINDOW DURATION FOR APOLLO LUNAR MISSIONS (Bellcomm, Inc.) 25 p

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MEMORANDUM FOR FILE

I. INTRODUCTION

It has been suggested that a reduction in the launch window for Apollo lunar landing missions could be beneficial. Three possible goals in reducing the launch window duration are:

1. Increasing the maximum launch vehicle payload capability.

2. Reducing the number of ships and aircraft required to support a lunar mission or conversely increasing the support capability of the existing fleet.

3. Reducing the complexity of the mission in terms of preplanning as well as actually carrying out the mission.

The reduction in the duration of the launch window is implemented by narrowing the useful launch azimuth sector from the current 26° sector (between 72° and 108°). The three goals mentioned above can each favor a change in the azimuth limits to different values. Therefore, the gains realized in attaining each of the goals must be weighed against the losses in terms of the other goals.

This memorandum presents the results of an analysis of the effects of reducing the launch window and presents information that is useful in determining which launch azimuth sector should be chosen. Sections II, III, IV, and V present the separate effects (payload variation, launch window duration, launch abort recovery forces, and communication and tracking effects) of changing the launch azimuth limits, while Section VI presents the combined effects of such a change.

This memorandum is not concerned with a change in the probability of an on-time and/or successful launch if the launch window is reduced.

II. PAYLOAD VARIATION

Since the 90° launch azimuth maximizes the launch vehicle payload capability (due to the earth's rotation), there is a payload loss associated with any launch azimuth other than 90°. Figure 1 illustrates the variation in payload loss with launch azimuth. The payload loss curve is approximately parabolic so that the payload loss increases in proportion to the square of the azimuth deviation from 90°. Thus, in terms of payload, the motivation for narrowing the launch azimuth sector is strong.

If it is desired to reduce the maximum payload loss by 500 pounds, from about 930 pounds to 430 pounds, the azimuth range from 78° to 102° (or even 103° since the payload loss curve is slightly unsymmetrical) could be utilized. If only 100 pounds of payload loss is to be tolerated (a gain of over 800 pounds compared to the present case), the azimuth range from 85° to 95° (or 96°) could still be used. Thus, once the maximum tolerable payload loss (compared to 90°) has been settled upon, the determination of the azimuth limits associated with this loss becomes a simple task.

The usefulness of this additional payload deserves further discussion. According to Figure 1, if the launch vehicle were capable of injecting 100,000 pounds on a translunar trajectory with a launch azimuth of 72°, it would be capable of injecting about 100,900 pounds if a 90° launch azimuth were employed. If the 72° and 108° launch azimuth limits are employed, the 100,900 pound payload capability is meaningless since the spacecraft fuel weights cannot be varied during the launch window. If it is known that the 90° launch azimuth is to be utilized, there is a potential payload gain of 900 pounds. The entire 900 pounds cannot be added to Command/ Service Module (CSM) inert weight because additional Service Propulsion System (SPS) fuel would be needed to supply the ΔV's for the various SPS burns. In fact, if one pound is added to the CSM inert weight, very close to one additional pound of SPS fuel must be added in order to provide the required ΔV. Thus, only about 450 pounds could be added to the CSM inert weight if the 90° azimuth were used. The question that remains is whether the SPS fuel tanks are large enough to hold the additional 450 pounds of propellant.

The quantity of SPS fuel required to carry out an operational "maximum SPS-fuel" mission was calculated to determine if certain missions would be limited due to insufficient SPS propellant tank capacity (the "tank-limited" case). The following procedure was used to calculate curves of SPS fuel required versus the injected spacecraft weight:

1. Assume a certain post-translunar injection weight (injected weight).

2. Subtract 3809 pounds for the Spacecraft-Lunar Module Adapter (SLA) panels and 381 pounds of translunar consumables.

3. Carry out SPS burns corresponding to 130 ft/sec. for dispersions and contingencies and a 3245 ft/sec. lunar orbit insertion (LOI).*

4. Subtract the 32,650 pound Lunar Module (LM), 589 pounds for two astronauts and 639 pounds of lunar orbit consumables.

5. Perform SPS burns corresponding to 797 ft/sec. for contingencies and inflight flexibilities and 4 ft/sec. for the CSM plane change.*

6. Add 689 pounds for the two astronauts and lunar returned samples.

7. Perform a 2734 ft/sec. SPS burn for TEI* and a 240 ft/sec. burn for dispersions and contingencies.

8. Add all of the fuel weights used for the SPS burns to determine the total SPS propellant required for this "maximum fuel" operational mission.

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*A scan was made of missions to candidate landing sites during 1969. Three launch dates per month were chosen on the basis of all Atlantic or all Pacific injections, at least a two day recycle time, good lunar lighting, and maximum fuel reserves. The 3245 ft/sec. LOI value, 4 ft/sec. CSM plane change value, and the 2734 ft/sec. transearth injection (TEI) value were found to be the combination requiring the maximum SPS fuel for missions to these "choice" sites, assuming a flight plan as described in Reference 1.

Figure 2 is a plot of the SPS fuel required for such a mission as a function of the injected weight. If the mission dependent ΔV's are to be changed while the injected weight is held constant, each additional ft/sec. of LOI burn uses about 4.7 pounds of the SPS fuel reserves and each additional ft/sec. of TEI burn uses about 2.8 pounds of the SPS fuel reserves.

If the usable capacity of the SPS tanks is about 39,700 pounds, then according to Figure 2, the injected weight can be as high as 103,000 pounds without compromising the assumed CSM ΔV budget. Under the assumptions made here, it is clear that the added 900 pounds of launch vehicle performance capability could be gainfully used since the currently reported performance (Reference 2) is below 103,000 pounds.

In other words, the spacecraft does not reach a "tank limited" condition, if choice lunar landing sites are used, until the injected weight reaches about 103,000 lbs. The use of a 90° launch azimuth would allow an inert weight increase of 450 pounds or an SPS propellant margin of 900 pounds.

III. LAUNCH WINDOW DURATION

It is fairly easy to show that the duration of the launch window associated with a given launch azimuth sector centered about 90° is virtually independent of lunar declination. The duration of launch windows associated with azimuth sectors centered on 90° is presented in Figure 3.*

The launch window situation becomes considerably more complicated when the azimuth sector is not centered on 90° (the motivation for not centering the azimuth sector on 90° will become more apparent in the next section) because the launch window duration becomes quite dependent on lunar declination.

Figure 4 presents the information required to determine the length of the launch window when the launch azimuth limits fall anywhere between 60° and 120°. As an example, using Figure 4, it is easy to see that if the azimuth sector from 90° to 100° is used and a Pacific injection is to be employed, the launch window will last about 1.9 hours if the lunar declination is -27°, but it will last only about 0.8 hours if the declination is +27°. Thus, pre-knowledge of the lunar declination (known

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*The duration of the launch window decreases slightly if the magnitude of the declination of the moon is greater than the geocentric launch site latitude of 28.45°. Such declinations occur briefly during many of the months from March 1968 to March 1970[3].

when the launch date is specified) can help determine the launch azimuth limits to be used on a given day.

IV. ABORT RECOVERY FORCES

Launch Aborts

Current planning for lunar missions indicates a need for 3 or 4 ships and 3 aircraft to provide adequate airborne pararescue access and ship retrieval capabilities in the Atlantic Ocean in the event of an abort during the launch phase (before the orbital insertion capability exists). The three aircraft are spaced along the ground track of the launch vehicle, their spacing dictated by their ability to fly to any point along the ground track. During the launch window the aircraft fly essentially southward to follow their assigned point on the ground track as the launch azimuth varies. Since the recovery aircraft generally can follow the motion of the ground track as the azimuth changes, the major factor determining the number of recovery aircraft is their ability to fly parallel to the ground track. Thus, the number of recovery aircraft is independent of the reduction in the launch window duration.

The situation concerning the recovery ships is more complicated because of their lower speed. By reducing the launch window to essentially zero, the possibility of having to travel over 500 n.mi. perpendicular to the ground track would be eliminated since the ground track would be known once the launch azimuth was determined. If 4 ships were to be used to cover a 26° azimuth sector between the 72° and 108° launch azimuth limits, 3 ships could provide essentially equivalent retrieval capability if the "launch-on-time" concept were used. If 3 ships were to be used with the 26° sector, 2 ships could not provide equivalent coverage if "launch-on-time" were used.

Earth Parking Orbit Aborts

When a finite launch window is employed, 3 ships and 3 aircraft are used to provide earth parking orbit abort recovery capability.* A ship and an aircraft are located at the point in the Pacific Ocean where all possible ground tracks for that orbit meet. Thus the longitudes and latitudes of the ships and aircraft are rigidly constrained in this case to three distinct points, one for each of the three possible earth orbits.

If the "launch-on-time" concept were used, the longitude constraint would no longer exist and the abort recovery areas could be moved east or west along the ground tracks. Moving two of the areas to the intersection of the two corresponding parking orbits eliminates the need for one ship and one aircraft since the two recovery areas would merge into one.

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*The current plan according to H. E. Granger, Landing & Recovery Division at MSC.

Thus, in terms of the entire abort recovery fleet, there is potentially a saving of two ships and one aircraft if the "launch-on-time" concept is used.

V. COMMUNICATIONS AND TRACKING COVERAGE

Although the present communications and tracking system was set-up to provide adequate coverage for any mission initiated with a launch azimuth within a 26° sector between 72° and 108°, there are naturally certain launch azimuths which provide better coverage than others. In addition, the size of the areas which must be covered by tracking ships and aircraft is reduced when the launch window is shortened.

Earth Parking Orbit (EPO) Insertion Coverage

Continuous tracking, telemetry, command and voice communications from liftoff through three minutes beyond insertion is currently specified as necessary(4) to backup and verify the onboard calculations of the navigation state vector. The choice of preferred launch azimuths in terms of launch and insertion coverage is complicated by the fact that the areas covered by land stations in this region are not yet fully defined due to uncertainties in multipath distortion effects at low elevation angles and the effects of local terrain and the antenna gimbal stops on the extent of coverage. A simplified coverage model can be used to give a conservative view of the contact available during the launch and insertion phases of the mission.

By assuming that no contact is obtainable for antenna elevation angles below 5°, the coverage circles in Figure 5 can be constructed. The notches or keyholes in the circles are caused by prelimits which prevent the antenna from pointing into conical regions (15° half angles assumed) north and south from the stations. Figure 5 shows that, according to the coverage model chosen, the southern Bermuda keyhole and the northern Antigua keyhole overlap yielding a region for which no coverage is available. Trajectories initiated with launch azimuths between about 90° and 104° would have a short period of time during the launch phase during which contact would be lost.

Several factors lead to the conclusion that the keyhole problem will not be as severe as indicated by the above analysis. First, data from the Manned Spacecraft Center(5) indicates that both of the Bermuda keyholes will have a half angle of 6.5° rather than 15°. In addition, the minimum elevation angle (determined by multipath distortion) for tracking by the Bermuda station may realistically turn out to be about 3° rather than 5° as assumed in Figure 5. (The minimum elevation angle for the Antigua station is a function of local terrain in addition to multipath distortion but in no case should the minimum antenna elevation angle be more than 5°.) If these possibilities are found to be true, the keyhole problem should no longer exist.

If the keyhole problem continues to exist, the choice of launch azimuths between 72° and 90° for the "launch-on-time" azimuth would avoid the problem altogether. It should be noted that if the discontinuity can be tolerated with the current finite launch window, it can be tolerated with the infinitesimal launch window associated with the "launch-on-time" concept. In this sense, there are no preferred launch azimuths in terms of communications and tracking during the powered phase of the launch.

A primary function of the three minutes of post-insertion tracking is to verify a safe perigee altitude for the earth parking orbit. Due to the locations of the Bermuda and Antigua stations, it is necessary to employ an insertion tracking ship in the Atlantic Ocean to provide much of the three minutes of post-insertion tracking coverage. Figure 5 indicates that the Antigua station provides some post-insertion tracking for certain launch azimuths greater than 90°. The actual post-insertion contact time(6) (above 5° and above 2° elevation) is shown in Figure 6.* More than one minute of contact time above 5° elevation is obtainable from the Antigua station for launch azimuths from 96° to 108°. Since the Antigua station could acquire the spacecraft well before insertion cutoff, there is no need to allow acquisition time for the post-insertion coverage.

Work done by Kaufman(7) at the Goddard Space Flight Center indicates that one minute of post-insertion ship tracking would give perigee uncertainties (1σ) of only about 1.7 n.mi. Kaufman points out that the 1.7 n.mi. figure is not the "worst" case uncertainty (due to dependence on the true anomaly at insertion) and that perigee errors are not Gaussian for near circular orbits. Thus, it is not strictly valid to say that,

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*In this case a 4° or 5° minimum antenna elevation angle is reasonable since local obstacles prevent line-of-sight contact below this value in certain directions around the Antigua station.

with 99.7% certainty (3σ Gaussian), the perigee can be predicted within 5.1 n.mi., but marked deviations from this value are not anticipated. Thus, it seems quite possible that early test missions will show that one minute of post-insertion tracking will suffice, especially if it is from a land station (since land stations are felt to give more accurate data).

It therefore seems likely that the insertion tracking ship could be eliminated completely if (a) approximately one minute of post-insertion tracking proves to be sufficient to assure safe perigee and (b) a launch azimuth between 96° and 108° is employed.

If it is decided to retain the insertion tracking ship, at least a 26° azimuth range can be provided with a minimum of 3 minutes of post-insertion coverage.

Earth Parking Orbit Coverage

The requirements[4] for communications and tracking coverage in earth parking orbit are:

(a) at least two 4-minute tracking, telemetry, command and voice contacts, at greater than 5° elevation, for each revolution prior to translunar injection (TLI).

(b) at least one 4-minute telemetry, command and voice contact, at greater than 5° elevation, between 90 and 30 minutes before TLI ignition.

If only Unified S-Band land stations are utilized to satisfy these requirements, certain slight deficiencies occur for launch azimuths between 72° and 77° and between 101° and 108°. None of these deficiencies could be considered severe in that the requirements would be satisfied for the 72° to 77° azimuth range if the 4-minute limit were changed to 3 minutes. The deficiency between 101° and 108° occurs because there are no 4-minute tracking passes after the United States is passed at the beginning of the third orbit. However, the Hawaii station does provide at least 2 minutes of tracking on the third parking orbit.

The inclusion of a tracking ship in the Indian Ocean, at 25°S latitude and 50°E longitude*, helps alleviate some of the above deficiencies. Figure 7 indicates the regions where

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*The general location of this ship is dictated by the need for post-TLI coverage (as discussed in the next section) but the specific 25°S, 50°E location was chosen to maximize EPO coverage.

EPO tracking coverage completely meets specifications and where there are slight violations both with and without the tracking ship.

If the launch window were reduced, the launch azimuth sector could be chosen to optimize earth parking orbit coverage but for no launch azimuth between 72° and 108° could the coverage be termed greatly deficient.

Apollo/Range Instrumentation Aircraft (A/RIA)

The A/RIA are used to monitor the time period from shortly before the translunar injection (TLI) burn ignition until shortly after TLI burn cutoff.* The ground track of the vehicle during this time period essentially follows the earth parking orbit ground track although altitude is gained during and after the burn.

The major factor which determines the geographical location of the TLI burn ground track on a given day is the lunar declination. On a given day the location of the injection burn is essentially fixed in inertial space so that the geographical location merely moves west at between 750 and 900 knots due to the rotation of the earth. This movement of the TLI burn ground track occurs whether the delay is due to time passage during the launch window (up to 4-1/2 hours or up to 4050 n.mi.) or due to the selection of the third orbit injection rather than the second (1-1/2 hours or up to 1350 n.mi.).

It is clear that utilization of the "launch-on-time" concept, which essentially reduces the launch window to a very short duration, would greatly reduce the geographical area to be covered by the A/RIA.

Much planning has been done based on the use of two A/RIA to cover a given TLI burn, with a total of six aircraft required to cover any possible burn on a given day. If the "launch-on-time" concept is used, three aircraft could form a line about 425 n.mi. west of the second revolution ground track (about 250 n.mi. perpendicular distance) to cover that possible burn. If the second revolution burn were scrubbed, the A/RIA could fly west about 500 n.mi. (400 knots, 1-1/4 hours) to a line about 425 n.mi. east of the third revolution burn ground track and cover that burn.

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*The current specification is for coverage from one minute before TLI ignition until three minutes after TLI burn completion.

The utilization of this scheme could reduce the required number of A/RIA from 8 (6 operational, 2 spares) to 4 (3 operational, 1 spare) but its use depends heavily on two factors:

(a) verification of the performance of the A/RIA communications system to confirm that the coverage from the assumed distances and elevations would be satisfactory (the maximum slant range assumed was 750 n.mi. and the minimum elevation angle was 5°).

(b) verification that the three aircraft have sufficient range and endurance to remain "on-station" for up to the maximum of about two hours for any possible TLI burn location through the month (a complex logistics problem).

An alternate scheme requiring an additional A/RIA (4 operational, 1 spare) would be to assign 2 aircraft to each of the two possible TLI burns. In this case a very short "on-station" capability would be required. In either case, if the "on-station" capability exceeded the requirement this excess capability could be used to provide TLI coverage for a finite launch window.

Although definitive answers are not possible due to the need for further test data, the potential does exist for reducing the number of A/RIA required to cover the TLI maneuver from 8 to as few as 4 if the launch window is drastically reduced.

Translunar Injection (TLI) Coverage

There is currently a requirement for continuous tracking, telemetry, and voice coverage (above 5° elevation) for at least 10 out of the first 20 minutes after TLI cutoff[4]. The coverage circles for the 14 Unified S-Band tracking stations corresponding to TLI-cutoff plus 10 minutes[8,9] show that all ground tracks receive coverage at that time except for an area in the western Indian Ocean and the eastern coast of southern Africa.* If all of the ground tracks were covered at TLI + 10 minutes, coverage would be assured at least until TLI + 20 minutes because the coverage circles expand rapidly as altitude is gained. The "uncovered" area can be covered by a single injection tracking ship in the Indian Ocean with a latitude of about 25°S and a longitude somewhere between 40° and 55°E (50°E longitude helps optimize EPO coverage). This single injection ship is required only for Atlantic injections but it is needed for all launch azimuths on certain days of the month (certain lunar declinations). Therefore, the employment of this single

*For translunar injection from the 2nd or 3rd parking orbit.

injection ship is not dependent upon the size of the launch window but only upon the lunar declination and the choice of Atlantic or Pacific injection opportunities.

Having discussed the manner in which a reduced launch window can be used to reduce the required number of support ships and airplanes, it must be pointed out that a more important gain from a reduced window may well be improved coverage of the TLI sequence with the existing fleet (particularly for the first lunar missions).

VI. ATTAINMENT OF GOALS

If the primary goal in reducing the launch window is to obtain the maximum launch vehicle payload (goal 1 in the Introduction) while still maintaining a finite launch window, Figures 1 and 3 can be combined to give the tradeoff between launch window and payload loss. Figure 8 is a plot of the payload loss for various launch window durations when the launch azimuth limits are symmetrical with respect to 90°.* A 1-hour launch window can be provided and the associated payload loss would be only about 50 pounds (a gain in useful launch vehicle payload of nearly 900 pounds, i.e., a CSM weight increase of about 450 pounds).

If a primary goal in reducing the launch window is a reduction in the number of ships and planes required to support the mission, the situation is more complicated. If less than about 1-3/4 minutes of post-insertion tracking is sufficient, it is possible to eliminate the insertion tracking ship. Figure 6 shows that any azimuth greater than 96° allows Antigua to track the spacecraft for at least 1 minute. If the 96° azimuth were utilized with a zero hour launch window ("launch-on-time") the payload loss compared to 90° would be about 80 pounds. Utilizing "launch-on-time" could permit the elimination of two abort recovery ships and one aircraft; in addition 4 A/RIA could could be eliminated.

If a finite launch window were to be used, the payload losses would increase rapidly with launch window duration. In this case it is impossible to construct a single graph of payload loss versus launch window duration because the launch window varies greatly with lunar declination. It is possible to determine the probability of a launch window of a certain duration

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*See footnote P. 4

by assuming that the moon's declination varies sinusoidally. Such plots are presented in Figure 9 for the azimuth ranges from 96° to 100°, 100° to 104° and 100° to 108°. The maximum payload losses for these three cases are 230 pounds, 460 pounds and 790 pounds respectively. (Note that there are actually short periods of time during 1968, 1969, and 1970 during which the magnitude of the declination of the moon is greater than the launch pad latitude. On such days, either the Atlantic or the Pacific injection opportunity may not exist if launch azimuth limits such as 100° and 104° are chosen.) The curve in Figure 9 for azimuth limits of 83° and 97° shows that a 2 hour launch window would be assured with the 83° and 97° limits. An insertion tracking ship would be required in this case.

The third goal to be considered in shortening the launch window is a reduction in the complexity of the mission in terms of preplanning and carrying out the mission. The satisfaction of this goal depends upon the size of the azimuth sector more than it depends upon the placement of the sector relative to the 90° azimuth case. In preplanning a lunar mission, a reduction in the magnitude of the launch azimuth sector would reduce the number of trajectories which would have to be investigated. The preplanning could generally be reduced in proportion to the reduction in launch azimuth span. The attitude timeline planning would perhaps be simplified since the orbital inclination would be more nearly constant. In addition, the logistics of properly placing the A/RIA would be simplified since the translunar injection burns occur in a more nearly constant geographical location.

VII. CONCLUSIONS

There are a number of effects resulting from a decrease in the duration of the launch window for Apollo lunar landing missions. Several conclusions regarding these effects are listed below.

(1) To maximize the launch window for a given payload loss, launch azimuth limits which are symmetrical with respect to 90° should be chosen.

(2) A one hour launch window can be provided with an associated maximum launch vehicle payload loss of about 50 pounds if the launch azimuth limits are changed to 86° and 94° (with 72° and 108° limits, the maximum payload loss is about 930 pounds).

(3) A one-half hour launch window can be provided with an associated maximum launch vehicle payload loss of about 20 pounds if the launch azimuth limits are changed to 88° and 92°.

(4) Any increases in launch vehicle payload capability achievable through a reduced launch window can be used to increase CSM margins, since the CSM does not reach a "tank-limited" condition until the injected weight approaches 103,000 pounds.

(5) If a launch azimuth of 96° is utilized, one minute of post-insertion tracking can be obtained from the Antigua station without using an insertion tracking ship and the launch vehicle payload loss compared to the 90° azimuth case will be about 80 pounds.

(6) Three launch abort recovery ships used with the "launch-on-time" concept provide capabilities equivalent to those provided by 4 ships if a 26° azimuth sector is used. An earth parking orbit abort recovery ship and an aircraft could be eliminated if the "launch-on-time" concept were utilized.

(7) There are no blatant violations of earth parking orbit communications and tracking requirements for any launch azimuth between 72° and 108°.

(8) Reducing the launch window essentially to zero provides the possibility of reducing the number of A/RIA from 8 to as few as 4 but such a reduction depends heavily upon the performance of the A/RIA communications equipment and upon the logistics of placing the aircraft at the appropriate locations.

(9) One tracking ship is required to satisfy post-translunar injection tracking requirements for <u>any</u> launch azimuth between 72° and 108° if the 2nd <u>or</u> 3rd orbit Atlantic injection opportunity is to be utilized. If the 2nd or 3rd orbit Pacific injection opportunity is to be utilized, no tracking ships are needed to satisfy the post-translunar injection tracking requirements.

T. B. Hoekstra

2013-TBH-srb T. B. Hoekstra

Attachments:
References
Figures 1-9

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2. "Weight and Performance Summary - November 1967 (U)", December 15, 1967, SE-015-005-1A, Prepared by Apollo Systems Engineering.

3. Hoekstra, T. B., "The Effects of Changing the Launch Azimuth Limits on the Launch Window, Launch Vehicle Payload, and Communications and Tracking Coverage - Case 310", Bellcomm Memorandum for File, April 6, 1967, p. 3.

4. "Apollo Saturn V Program Support Requirements Document", (PSRD), revision issued 10/2/67, p. 2020.5.

5. Minutes from the 24th Lunar Landing Flight Operations Plan ("G" Mission) Meeting, December 13, 1967, Enclosure 8.

6. Data received by telephone from Robert H. Benson/R-AERO-FT at MSFC.

7. Kaufman, Bernard, "Monte Carlo Simulation for Apollo Go, No-Go Verification", NASA X-507-67-61 (GSFC) February 1967, Figure 5.

8. Taylor, James J., "Translunar Injection Positions for 1968 and 1969", MSC Internal Note No. 66FM-42, April 20, 1966, Figure 9.

9. "Injection Ship Positioning Technique for Apollo Lunar Missions", TRW Note No. 67-FMT-523 Project Apollo Task MSC/TRW A-134, 23 June 1967, Figures 5 and 6.

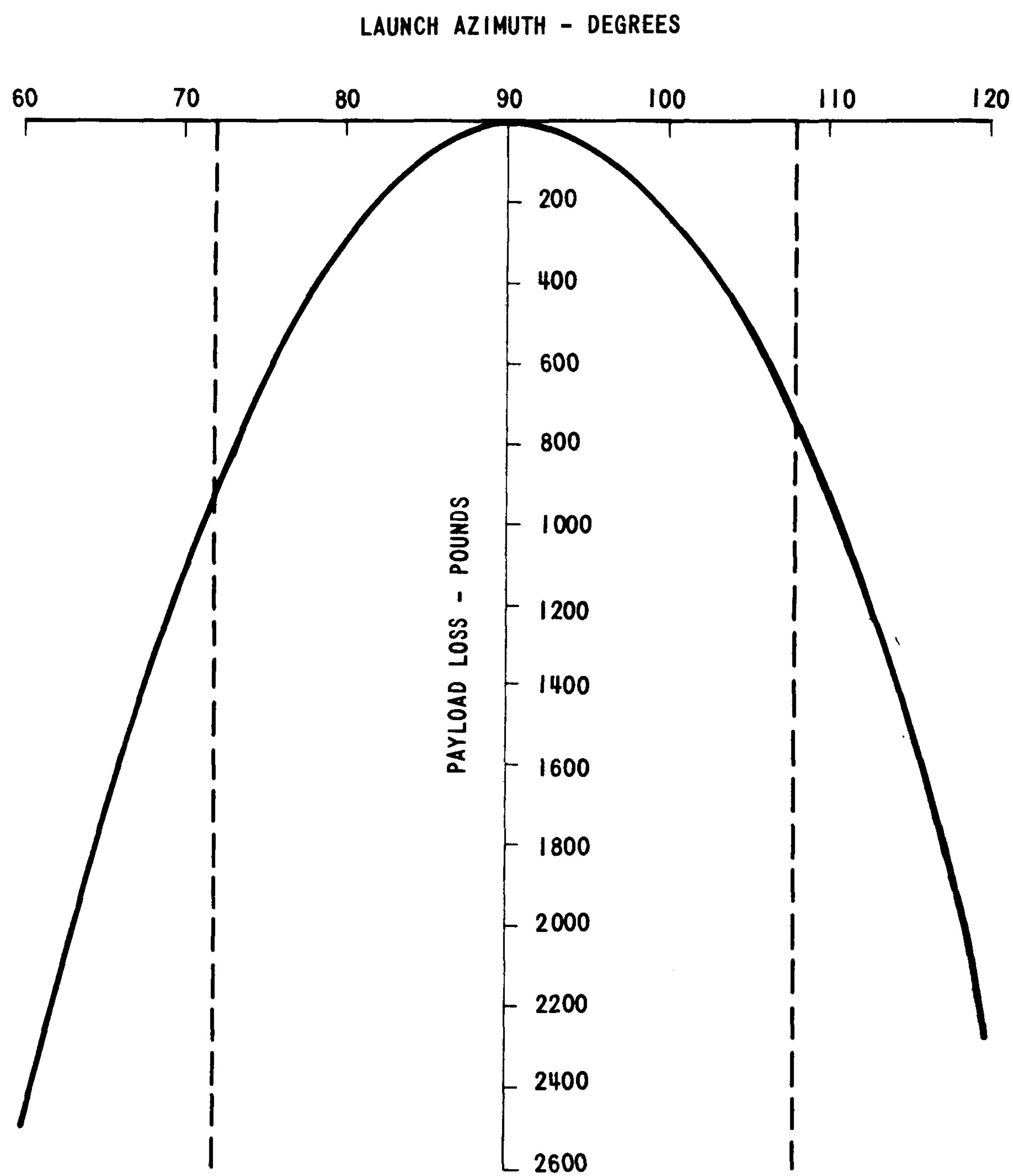


FIGURE 1 - LAUNCH VEHICLE PAYLOAD LOSS FOR VARIOUS LAUNCH AZIMUTHS

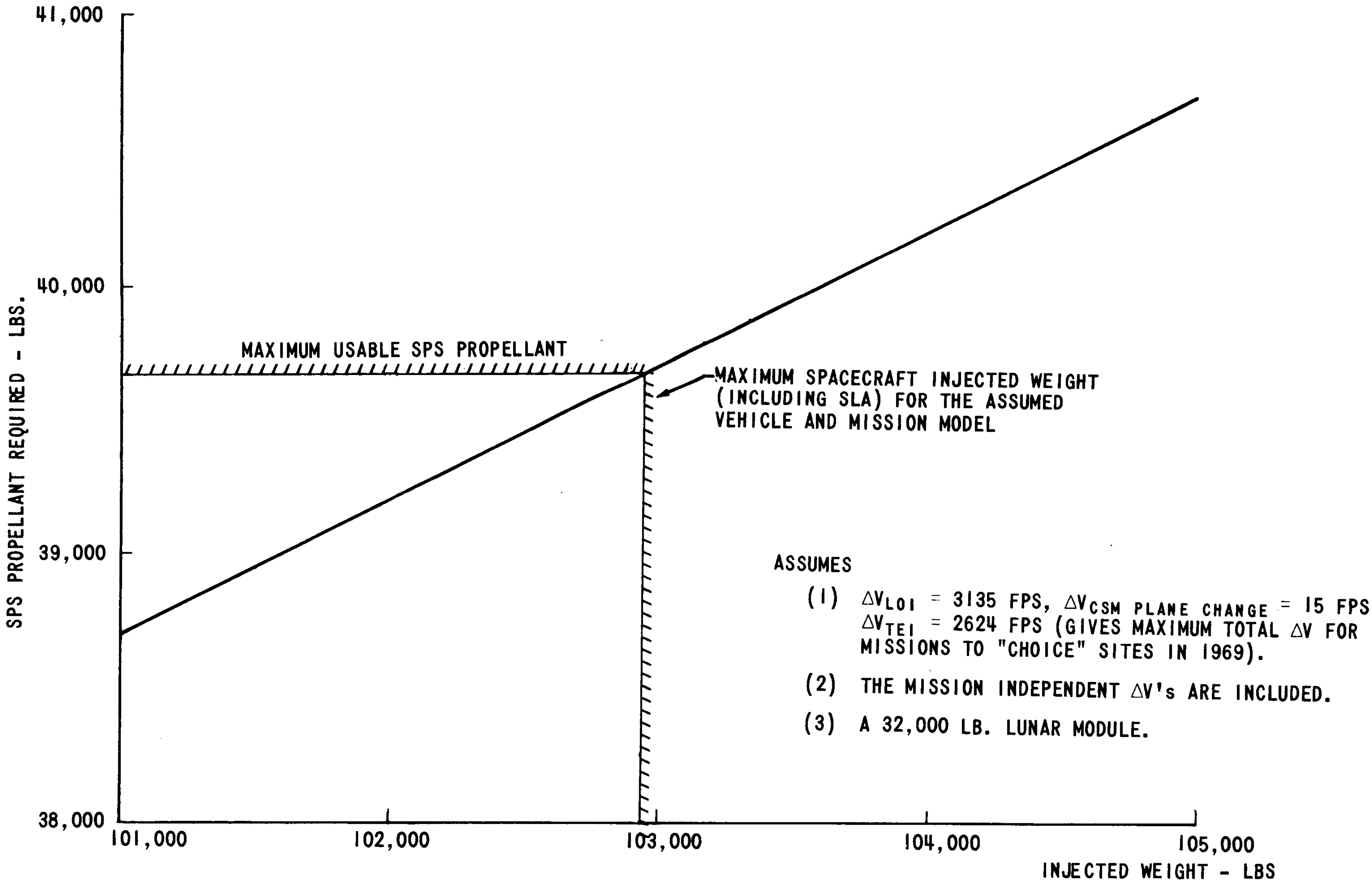


FIGURE 2 - SPS PROPELLANT REQUIRED FOR VARIOUS SPACECRAFT INJECTED WEIGHTS

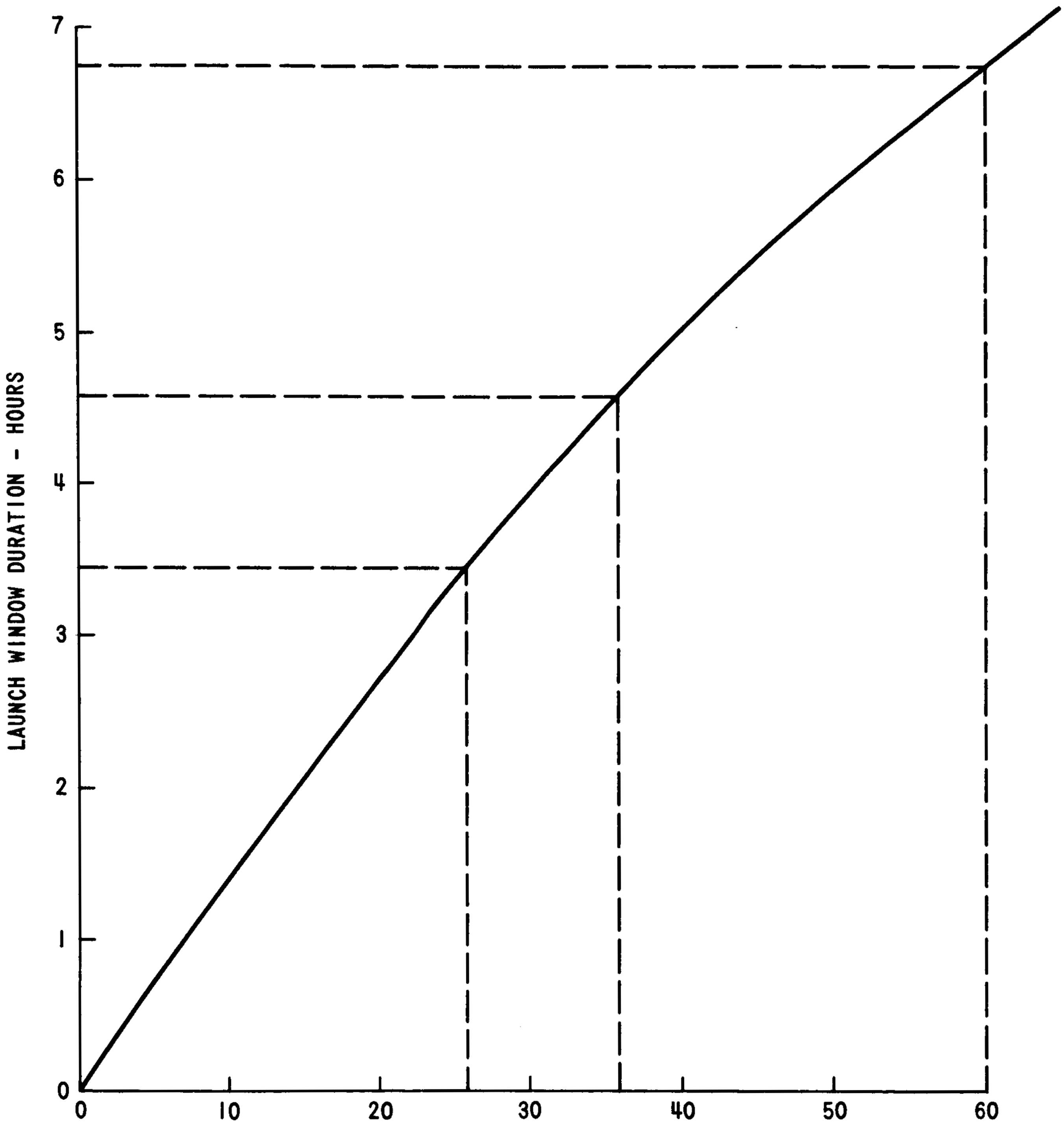


FIGURE 3 LAUNCH WINDOW DURATION FOR VARIOUS LAUNCH AZIMUTH SECTORS (AZIMUTH SPREAD CENTERED ON 90°)

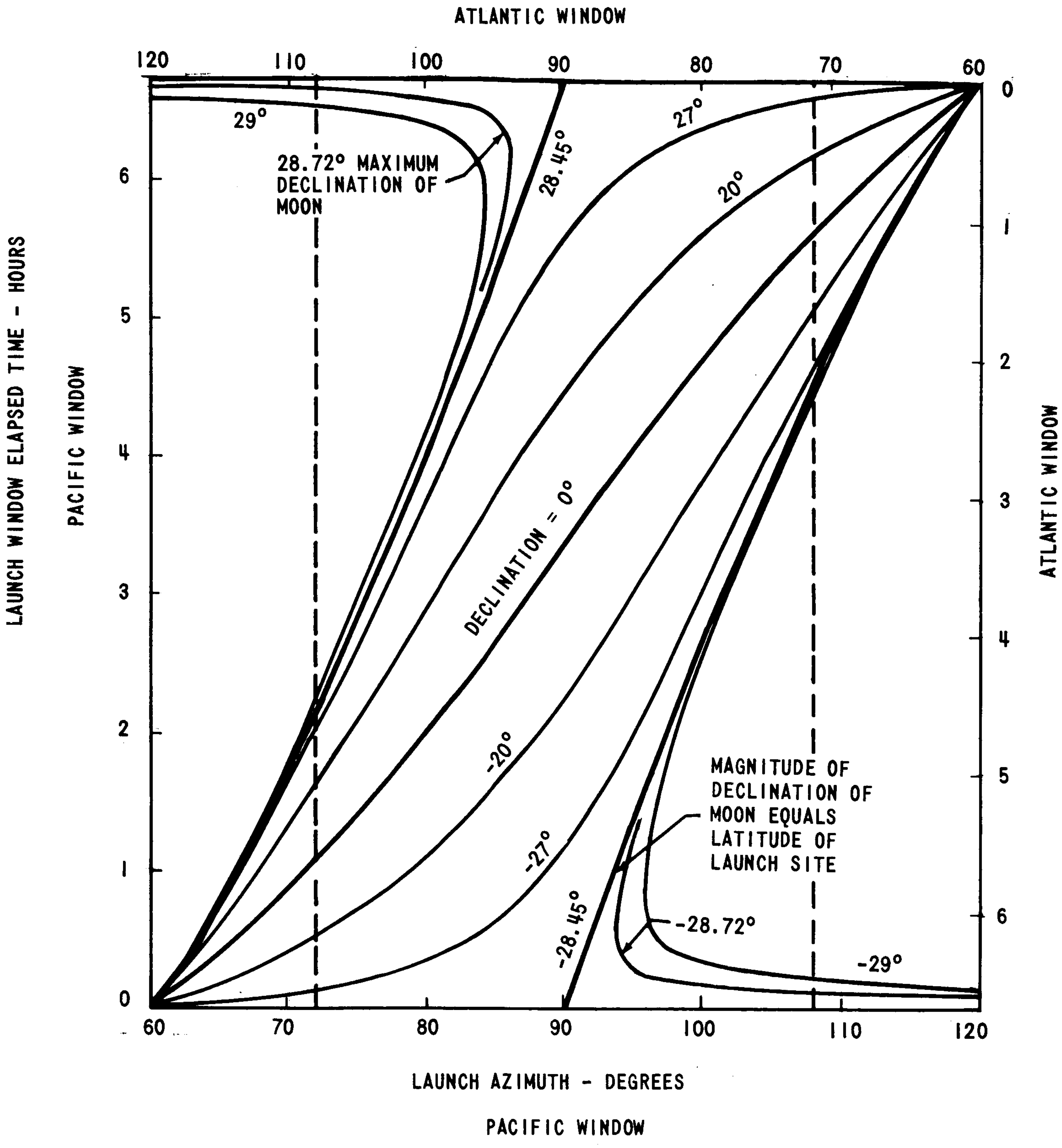


FIGURE 4 - LAUNCH WINDOW ELAPSED TIME VS. LAUNCH AZIMUTH

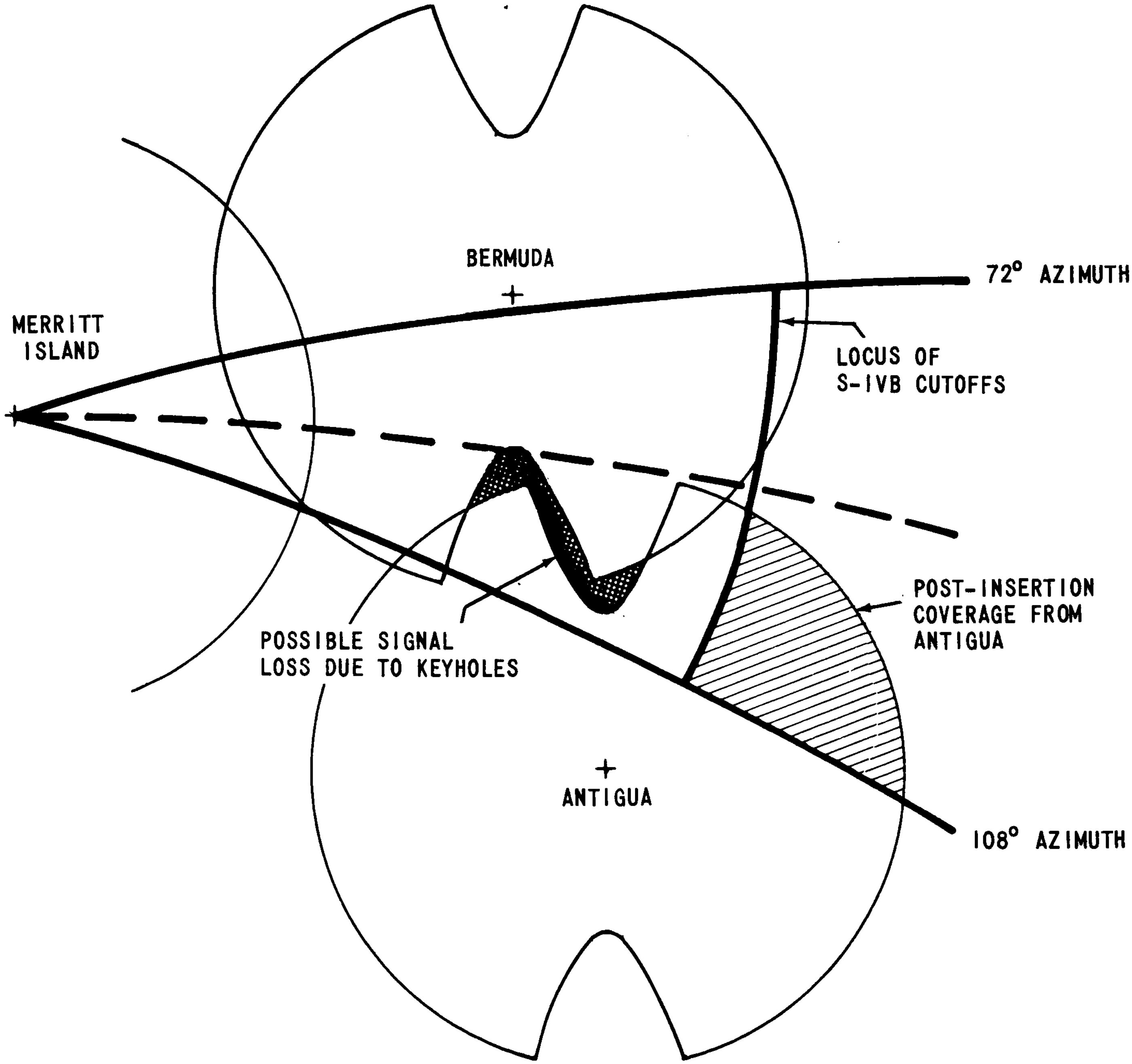


FIGURE 5 - SCHEMATIC VIEW OF INSERTION TRACKING COVERAGE FROM LAND STATIONS
(COVERAGE CIRCLES SHOWN FOR 5° ELEVATION ANGLES AND 15° KEYHOLES)

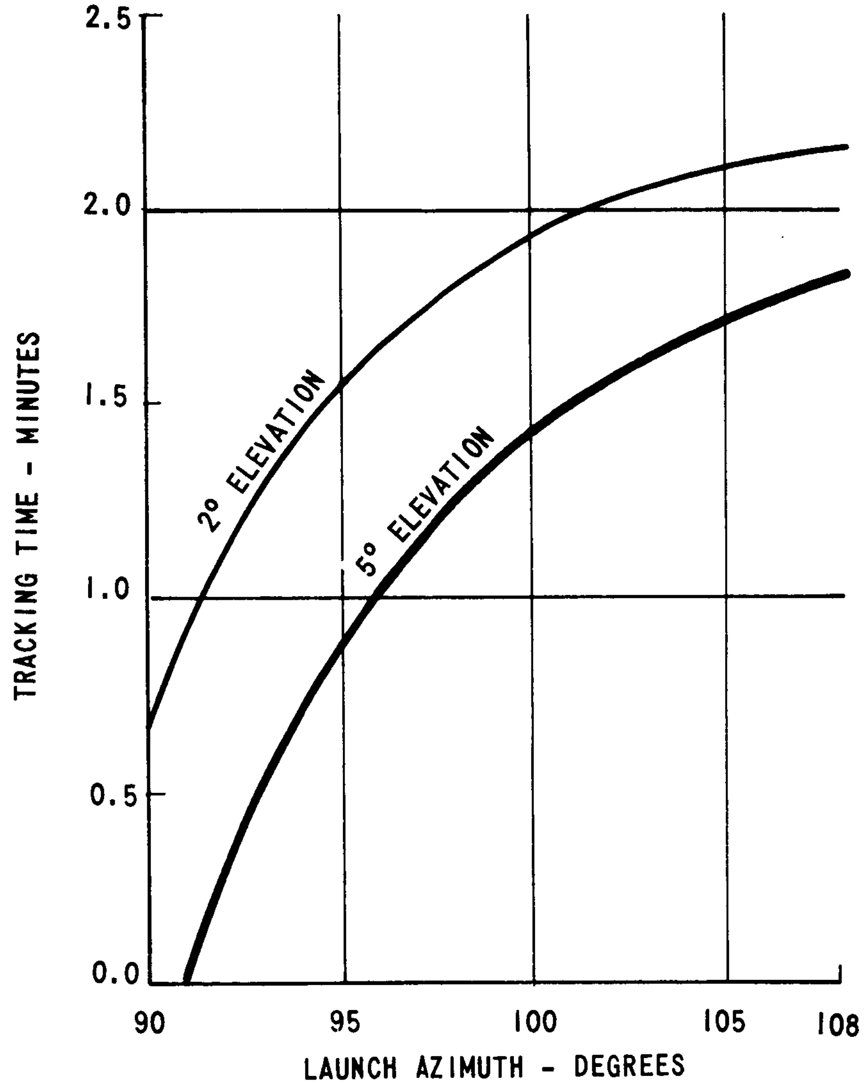
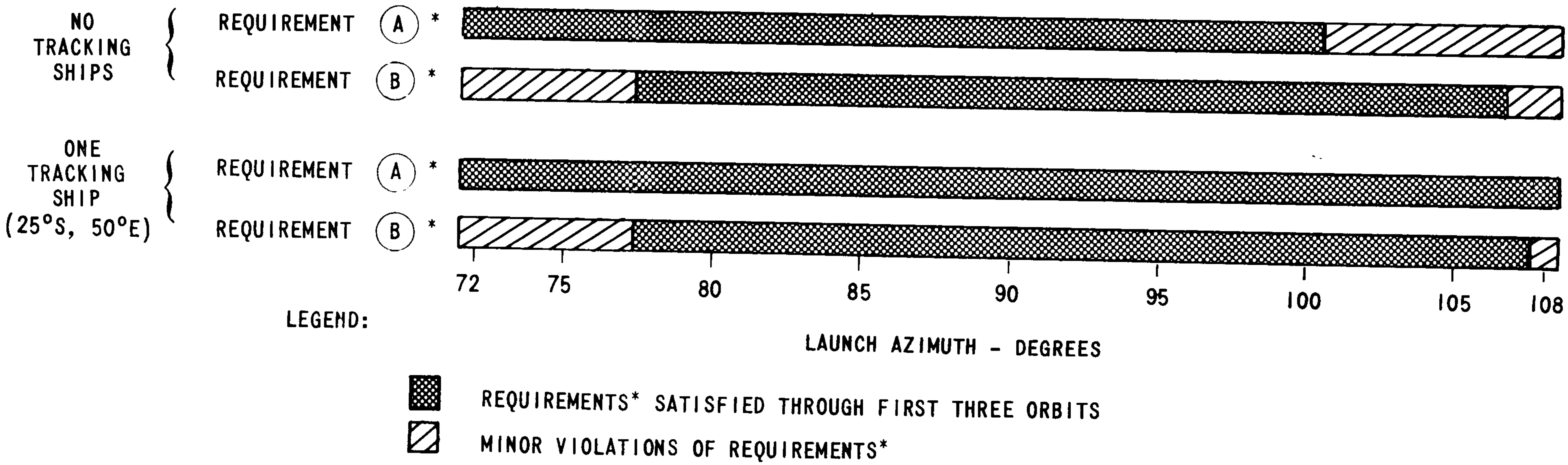


FIGURE 6 - POST-INSERTION TRACKING TIME FROM ANTIGUA



*REQUIREMENTS

(A) 2 UNIFIED S-BAND CONTACTS > 4 MINUTES DURATION, > 5° ELEVATION PER REVOLUTION (ASSUMED EVERY 93 MINUTE SEGMENT HAD TO CONTAIN 2 PASSES)

(B) 1 UNIFIED S-BAND CONTACT > 4 MINUTES DURATION, > 5° ELEVATION BETWEEN 90 AND 30 MINUTES BEFORE TRANSLUNAR INJECTION

FIGURE 7 - SUMMARY OF SATISFACTION OF EARTH PARKING ORBIT COMMUNICATIONS REQUIREMENTS* FOR VARIOUS LAUNCH AZIMUTHS

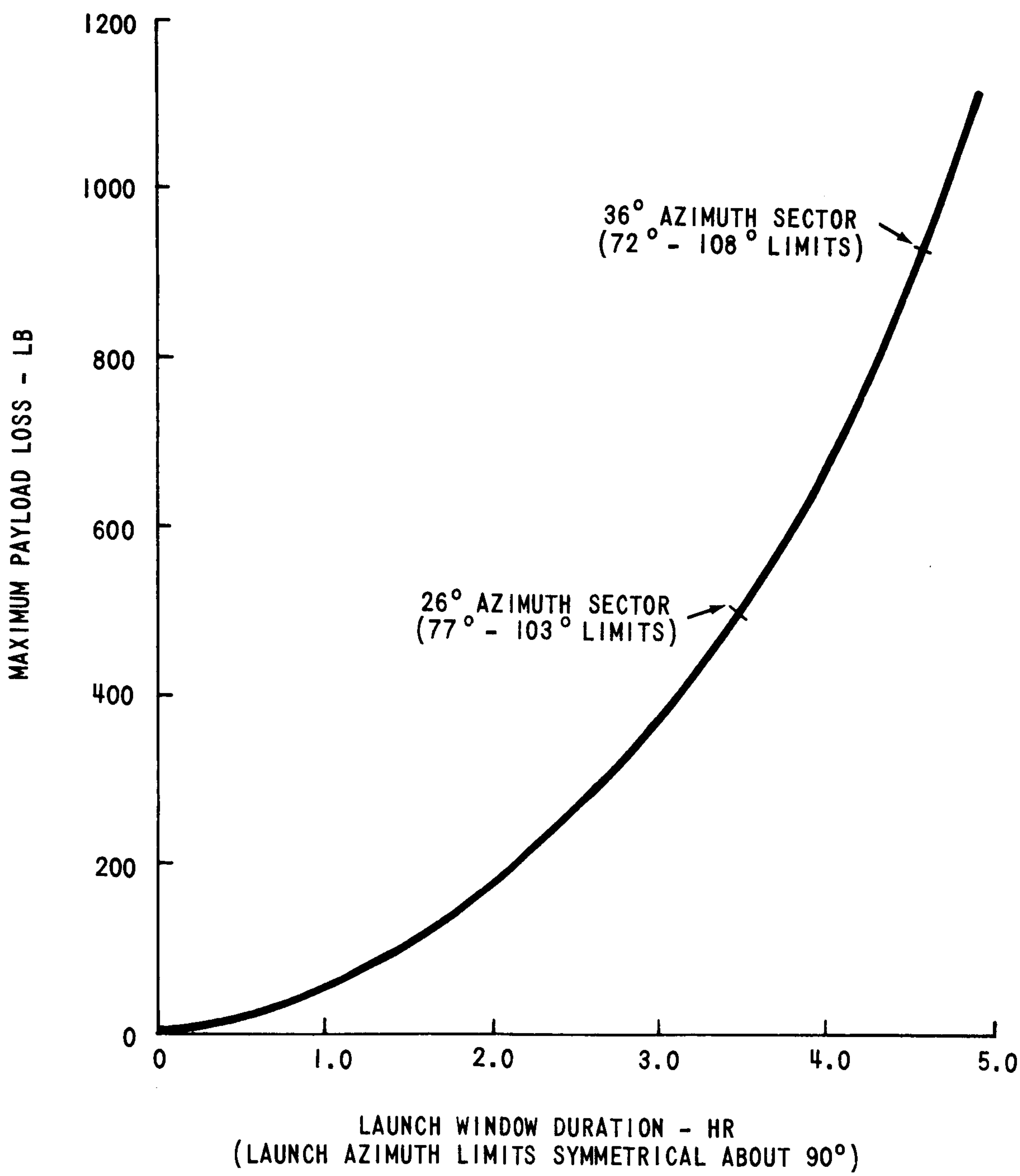


FIGURE 8 - MAXIMUM PAYLOAD LOSSES FOR VARIOUS LAUNCH WINDOW DURATIONS (LAUNCH AZIMUTH LIMITS SYMMETRICAL ABOUT 90°)

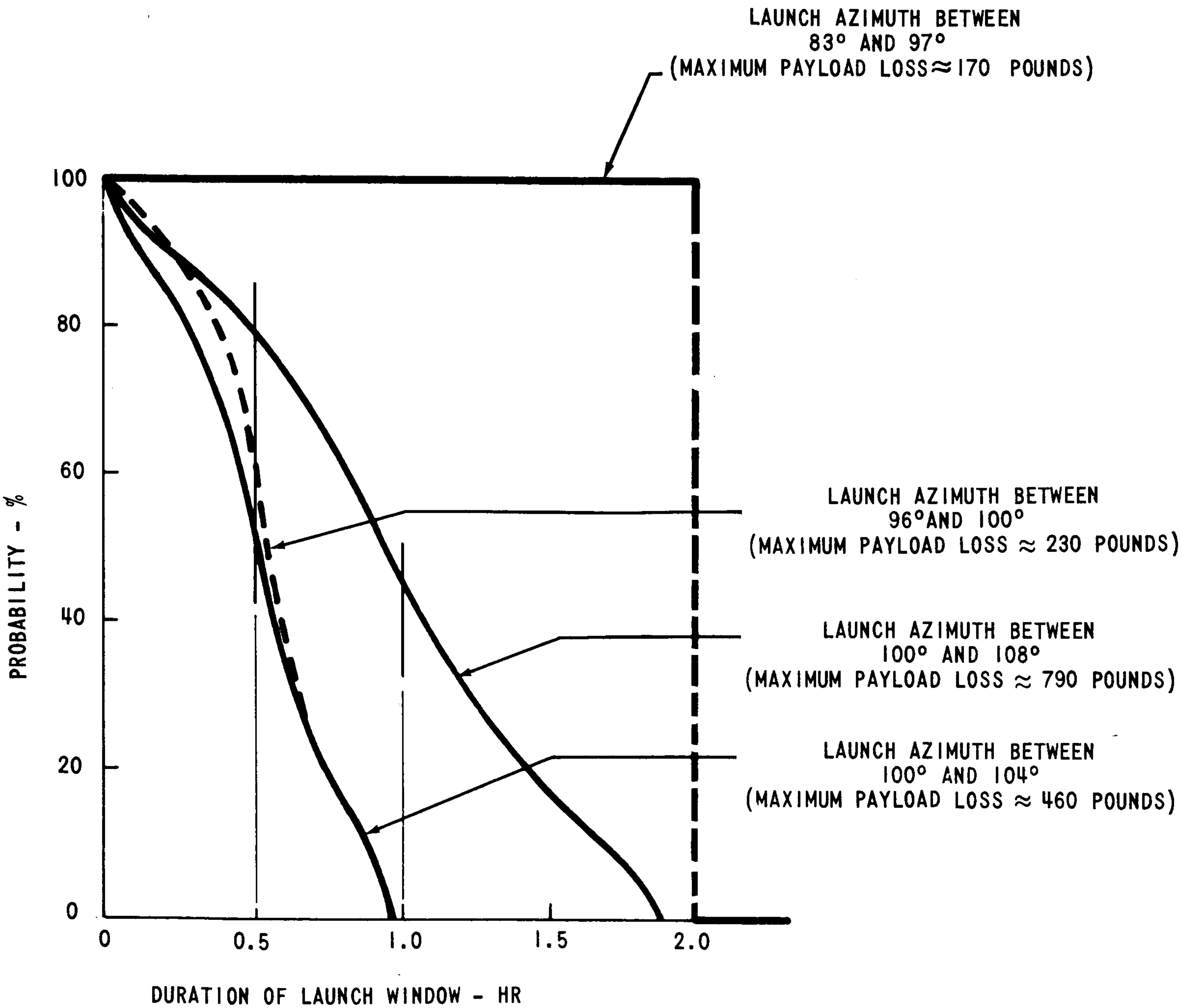


FIGURE 9 - PROBABILITIES OF LAUNCH WINDOWS OF VARIOUS DURATIONS
(1968 - 1970 TIME PERIOD; LUNAR ORBIT INCLINATION = 28.45°)

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